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A Study of Solar Thermal Propulsion System Enhancement Via Thermal Storage and Thermal-electric Conversion

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ABSTRACT

Solar thermal propulsion (STP) systems optimized for microsatellites offer thrust and specific impulse combinations that exceed the capabilities of similarly sized, conventional chemical and electrothermal rockets. Past research into STP has indicated these advantages, with the caveats that propulsion can only be augmented during insolation and that a thermal power system competes with the traditional electric power systems required to support other spacecraft components. An STP system augmented with thermal storage and thermal-to-electric conversion is suggested and analyzed as a means to bypass these drawbacks. Targeting a temperature level of 1500-2500 K via solar concentration, such a system can yield a high-performance microsatellite with superior thrust and specific impulse (Isp) levels. It is shown that a phase-change thermal storage material can potentially be used to significantly surpass the energy density of traditional battery-powered systems. Coupled to a thermal photovoltaic system for electrical power, such a system can provide the advantages of STP along with meeting the requirements of traditional satellite payloads. Combined with the other efficiency advantages offered by STP, such a system has the potential to significantly enhance the capabilities of microsatellite systems while reducing mass and launch costs or increasing payload fractions. It is shown that individual technologies and materials exist for creating all required subsystems, including solar collection and concentration, thermal storage, insulation, heat exchange, and electric conversion. No significant breakthroughs are required to fully design a satellite based on the suggested technology; ensuring material compatibilities and optimal subsystem integration are the most notable challenges that must be overcome.

INTRODUCTION

A recent review of propulsion technologies[1] has indicated the strong potential performance offered by Solar Thermal Propulsion (STP) in microsatellite systems. Analysis shows that STP systems currently in development, with specific impulse values of 400-1000 seconds, can provide a propulsion system mass fraction (including propellant) of less than 50% with a 1.5 km/s velocity increment capability. With microsatellite STP thrust levels of 1 N, the entire velocity increment could be delivered in less than one day, providing the potential for a highly capable spacecraft.[1]

Traditionally, however, STP has been viewed as somewhat limited due to the requirement for solar illumination of the collector during times when propulsion is needed. Proposed fiber optic collection of concentrated sunlight can yield some extra flexibility in terms of spacecraft attitude and positioning by decoupling precisely-aimed collectors from the thruster and other [physical2-6]. Combining STP with a means of thermal storage, however, could vastly increase the utility of the propulsion system for a variety of missions; adequate thermal storage could potentially yield a spacecraft that can produce thrust on-

demand even when in eclipse, offering the flexibility of a conventional chemical or electrothermal system with the superior performance and robustness of an STP system. Further, a thermal storage system could yield benefits in terms of augmented thrust relative to a similarly sized non-storage STP system if both direct-gain solar illumination and stored thermal energy can be used simultaneously; this sort of design could likewise be used to decrease the size of the overall solar concentration system for a spacecraft with only intermittent propulsion requirements.[2]

An additional perceived disadvantage of an STP system is the need for large solar concentrators. These collectors traditionally require a trade-off with traditional power systems for mass and space onboard a satellite system. It is suggested here, however, that the use of an effective thermal storage system, combined with a means to convert thermal energy to on-demand electrical power, could eliminate the need for photovoltaic systems and accompanying batteries.

This paper will analyze the prospects for utilizing a solar thermal system with thermal storage as the primary energy source and propulsive device on board a satellite. This study will not offer a complete design of any particular subsystem, but will seek to survey each area of technological interest for such a system. The ultimate goal will be to determine the technological developments required to produce a system that offers notable cost, efficiency, and performance advantages over a conventionally powered satellite. Solar collection and concentration, thermal storage materials, thermal insulation, heat transfer to the propellant, and the conversion of stored thermal energy to electricity will be discussed.

STORAGE SYSTEM GOALS

Previous studies have examined the prospect of STP with thermal storage[2,3], including the Solar Orbit Transfer Vehicle, which included a means of thermal electric generation[7-10]. Key differences to be explored here are the optimization of thermal storage using a phase change material, design considerations assuming a microsatellite system in low Earth orbit, and practical considerations in terms of propellant and technology selection. With this in mind, the target will be to devise a system that competes favorably against traditional chemical and electrothermal rockets coupled to battery-based electrical systems. The goal is not a full system design, but a determination of current capabilities and an analysis of technological developments (in terms of material properties and system designs) required to make such a system feasible and competitive.

Because a satellite will reliably pass from eclipse into illumination, thermal energy need not be stored indefinitely. Likewise, it is expected that a thermal storage medium will not have significant life-limiting problems associated with repeated heating and cooling cycles; on the contrary, systems that require the use of batteries may need to be significantly oversized to limit the degree of discharge that occurs during each loss of power from photovoltaic systems. Therefore, while the target of the storage system will be to provide for a specific energy density (kJ/kg) that can compete with batteries, the notion that state-of-the-art batteries may need to have a significantly larger total storage capacity should be kept in mind.

Taking round numbers for a LEO orbit, it is assumed here that a typical satellite might have a 100-minute orbit with 40 minutes in eclipse. To compete with the power numbers of a typical 100 kg microsatellite, a power output of 100 Watts[11] during eclipse is suggested for the thermal-storage system. After accounting for efficiency and losses in the storage and electrical conversion systems, 67 Watt hours (roughly 240 kJ) would therefore need to be recovered during each eclipse. Assuming that an overall efficiency of 25-50% can be achieved for the insulation of the thermal storage medium and the thermal-electric conversion system, then perhaps a total thermal storage capacity of 500-1000 kJ is required on board the spacecraft. As this stored energy could potentially be used for electrical power generation or for augmented thrust during eclipse or insolation, it may be worth considering implementing a design with superior performance that can simultaneously provide 100 W of power to an electrical system and

additional power to the STP system; whether or not this is practical will depend on system efficiencies and the mass (i.e.: heat capacity) of the thermal storage system required.

PROPULSION REQUIREMENTS

In order to provide a propulsion system that not only competes with existing flight-ready technologies, but proves to provide significant enough advantages that a full redesign of the satellite power system is justified, we must demand high performance of an STP system. The primary competitors for the high-thrust, mid-Isp capabilities of an STP system are in-space chemical and electrothermal rockets. In order to compete with these systems, an STP system must provide similar or greater thrust with a notable increase in Isp. It has been shown in the literature that an STP system sized for a microsatellite is capable of producing a relatively high thrust level of 1 N[1,5]. In order to surpass the best chemical systems available for microsatellites, a target Isp in the range of 300-400 s is the goal, and is certainly achievable[1,3,5].

The primary propellant options discussed in the literature for STP systems are hydrogen and ammonia. Hydrogen can offer Isp values on the order of 1000 s, but is not space-storable[2]. Ammonia, on the other hand, is readily storable in space, and several studies show that with a pre-nozzle temperature above 2500 K, Isp values of 407 s can be achieved[2,5]. Ammonia propellant also offers additional advantages in terms of system simplicity: the vapor pressure of the ammonia propellant can be used to pressurize the propellant and move it from the storage tank to the thruster[2], so an active feed system is not required. Ammonia thermally dissociates into hydrogen and nitrogen at high temperatures; this provides lighter exhaust species that can be ejected at higher velocity with the simultaneous disadvantage of the absorption of significant heat when bonds are broken[4].

Theoretically, the exhaust velocity and the Isp can be calculated as:

$$V_e = I_{sp} g = \sqrt{\frac{T k_B}{m} \frac{2\gamma}{\gamma - 1}\left[1 - \left(\frac{P_e}{P}\right)^{\frac{\gamma-1}{\gamma}}\right]}, \qquad (1)$$

where V_e is the exhaust velocity, T and P are the upstream propellant temperature and pressure, respectively, P_e is the exit pressure from the nozzle, m is the average propellant particle mass, γ is the ratio of specific heats for the propellant, g is the gravitational constant, and k_B is Boltzmann's constant.

Direct calculation of the exhaust velocity from temperature alone requires assumptions of the upstream pressure, the degree of disassociation of ammonia, and calculations based on the nozzle area ratio to determine the exit pressure. Additionally, even with the assumption that ammonia does not disassociate, the ratio of specific heats, γ, will vary with temperature. At low temperatures, in which only the translational and rotational modes of molecular motion are activated, γ is given as 1.31. As temperature increases and the 6 vibrational modes are activated, the ratio of specific heats will decrease, yielding a γ value closer to 1.1.

Therefore, for this analysis, we will choose a basis value that has been noted in the literature: a temperature of 2500 K for ammonia should yield an Isp of 407 s in a typical solar thermal rocket[2,5]. For estimating performance with varying temperature, a rough scaling with the square root of T is utilized; the result is displayed in Figure 1. Note that the estimated performance curve closely approximates predicted and measured performance of solar thermal rockets under development.

Therefore, throughout the rest of this paper, the analysis will focus on the requirements associated with concentrating solar flux onto a thermal storage medium in order to achieve something near the 2500 K target temperature. This requires a solar concentration ratio of 10,000:1 [2,10], a level of concentration that has been demonstrated experimentally[4]. The thermal storage medium must be able to adequately store energy in this temperature regime, and insulators must be able to maintain that

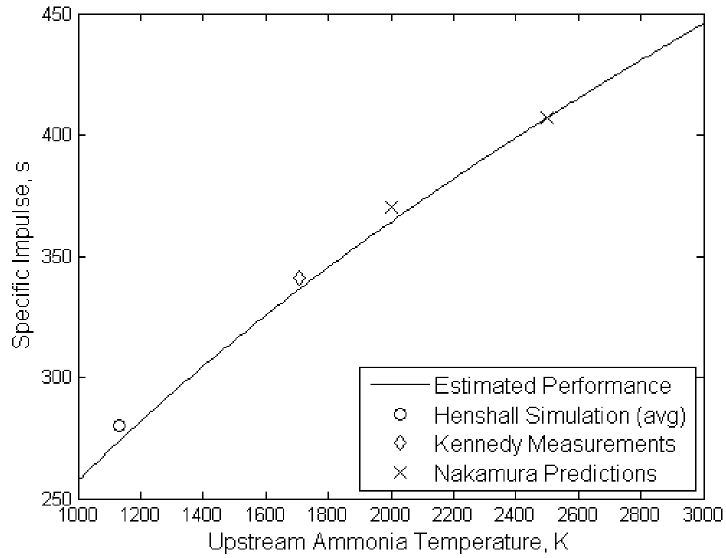


Figure 1: Isp vs. upstream temperature for an ammonia-fueled solar thermal rocket.

temperature for a significant amount of time during eclipse. Further, a heat exchanger for transferring energy from the thermal storage medium to the propellant will need to be developed, and a means of thermal-electric conversion will need to be devised. All materials in contact with the thermal storage medium must, of course, tolerate the highest temperatures achieved by the thermal storage medium without significant chemical or physical changes.

THERMAL COLLECTION

The collection of thermal energy from the sun is essentially a solved problem, with collectors appropriately designed for in-space use that achieve concentration ratios over 10,000:1 in ground-based experiments[4]. There is significant confidence that an adequate solar concentration and collection system can be designed to produce temperatures over 3000 K, and that the target temperature of 2500 K can certainly be achieved[5].

Variations of solar concentration and collection exist in currently developed STP designs, in ground-based solar thermal power generation, and even in concentrated photovoltaic systems with both terrestrial and space-based applications. Proposed designs for propulsion systems include parabolic, spherical, and fresnel concentrators used to focus sunlight by orders of magnitude onto a small surface[2,4]. For light-weight, imperfect primary concentrators, small secondary concentrators (with high temperature handling capabilities) can be added to more precisely focus the collected light[2]. Concentrator designs include inflatable and deployable systems as well as light-weight rigid mirrors[2,4,5,7-10]; concentrator masses below 1 kg/m^2 are common[2], and concentrator mass will certainly compete well with that of a photovoltaic (PV) system. Due to the higher efficiency of solar reflection compared to photoelectric conversion, the size of a concentrating system can be notably smaller. The use of fiber optic lines to couple together the collected light from multiple small concentrators can also be used to further decrease the mass of an STP system as smaller concentrators may have a much lighter mass for a given surface area[2-4].

It is expected that 72% efficiency of the entire solar collection system can be achieved[5]; this includes inefficiencies associated with reflecting and concentrating the incident light, collecting and transmitting that light along a fiber optic cable, and delivering it to the thermal storage medium[5]. Designed concentrators can achieve 3300 K or higher as shown by Nakamura, et al.[5]. It should be noted, however,

that current engineering models are only 35% efficient. Even at this level, the efficiency may be compared with that of triple-junction PV panels for use in space, although the "downstream" efficiencies of a solar thermal system will likely be considerably less than that of the simple batteries and power supplies that might be used to handle PV-generated power. However, Nakamura[5] indicates that there are a few simple upgrades to the basic engineering systems that would double the overall efficiency. For example, a relative efficiency increase of 9% will be achieved by simply operating the solar concentrators in space where conditions of the air will not affect the path of light[5]. Additionally, Nakamura, et. al, state that in laboratory models, a concentrator with a reflectivity of 80% was used; reflectivity above 90% requires only careful engineering[5]. Targeting these and a few other minor improvements to the design, researchers expect overall efficiencies over 70% for a fully space-qualified collection system. There is, however, some disagreement as to the absolute maximum that can be achieved: Henshall, et. al, for example, places the fiber optic transmission efficiency at a maximum of 80%[2,3], while Nakamura suggests that 90% is readily achievable[5]. Significant improvements over the 35% starting point, however, is certainly likely as a system is developed and optimized for use in space[5]. With the expected improvements in optical system efficiency achieved, well designed storage, heat transfer, and electrical conversion subsystems should be able to achieve a marked step forward relative to traditional PV-battery systems.

THERMAL STORAGE MEDIA

The thermal storage medium (TSM) must offer a combination of several traits in order to fulfill the needs of the satellite. It must be able to store energy that competes on a per-mass basis with traditional battery-based electric storage systems; this equates to roughly 500 kJ/kg. The TSM must handle the temperatures required without undergoing significant chemical or volumetric changes over time, it must not corrode or otherwise damage materials that must contact it, and it should be non-radioactive and relatively non-toxic. Ideally, for the propulsion subsystem of the spacecraft to offer easily predicted performance, the TSM should change phase (i.e.: solid to liquid) near the desired operating temperature for the propellant; the heat of fusion offers a significant energy storage capacity relative to simply heating a given material[12], and constant temperature operation would allow for an easily predicted thrust level from the propulsion system. It is suggested that a phase-change TSM could either be housed within a simple vessel, or it could be contained within the pores of a solid matrix material that will not melt at pertinent temperatures[12,14]; this second approach is commonly studied for terrestrial applications and is useful if it is necessary to maintain the shape of the thermal storage medium, and may also be used to enhance the heat transfer rate to and from a non-conductive phase change material[12,14].

Traditional TSMs intended for ground-based systems are not ideal for an STP system as they have not been targeted for the high-temperature operation required for propulsion. The traditional TSMs with the highest phase change temperatures are listed in Table 1.

Table 1: Traditional phase-change thermal storage media with highest melting points[13].

Compound	Melting Temp [K]	Heat of fusion [kJ/kg]
$MgCl_2$	987	452
45.8%LiF + 54.2%MgF_2	1019	Not Available
NaCl	1073	492
53.6%NaF + 28.6%MgF_2 + 17.8%KF	1082	Not Available
66.9%NaF + 33.1%MgF_2	1105	Not Available
Na2CO3	1127	275.7
Salt Ceramics $NaCO_3$-$BaCO_3$/MgO	773-1123	415.4
KF	1130	452
K2CO_3	1170	235.8

Table 2: Candidate non-radioactive thermal storage media with melting points between 1500 and 3000 K [15]; some property and cost figures have been gleaned from internet sources and supplier specification lists and direct references are not provided here. Thermal conductivities are noted at room temperature.

Element/ Compound	Atomic Number	Melting Temp [K]	Heat of fusion [kJ/kg]	Thermal Conductivity [W/mK]	Cost [/kg]
Manganese	25	1519	235	7.8	$17
Beryllium	4	1560	1312	200	$800
Silicon	14	1687	1785	149	$3.20
Nickel	28	1728	298	90.9	$15
Cobalt	27	1768	272	100	$210
Yttrium	39	1799	128	17.2	$2200
Iron	26	1811	247	80.4	$67
Scandium	21	1814	313	15.8	$14,000
Palladium	46	1828	157	71.8	$10,600
SiO_2	N/A	1923	188	~1	
Lutetium	71	1925	126	16.4	$10,000
Titanium	22	1941	295	21.9	$61
Zirconium	40	2128	153	22.7	$160
Si_3N_4	N/A	2173 (decomposes)		30	
Chromium	24	2180	403	93.9	$200
Vanadium	23	2183	422	30.7	$2,200
Rhodium	45	2237	258	150	$110,000
$MoSi_2$	N/A	2303			
Boron	5	2350	4600	27.4	$5000
Hafnium	72	2506	152	23.2	$1,200
Ruthenium	44	2607	381	117	$14,000
Iridium	77	2739	213	147	$42,000
Niobium	41	2750	323	53.7	$170
Molybdenum	42	2896	390	138	$100

The materials listed in Table 1 do not achieve the phase-change temperature that would be ideal (2500 K) for an ammonia-fed STP system. It is noteworthy, however, that the heats of fusion of several of these materials are in the ballpark of what is available for the specific energy density of lithium ion batteries (~500 kJ/kg). Additionally, even at temperatures of only 1000 K which can be achieved by several traditional phase-change TSM's, the Isp produced could exceed 250 s, which would be very competitive with chemical rockets for microsatellite systems. Since the goal here is to provide a more significant performance enhancement, however, it may be necessary to seek out higher temperature materials.

There are a variety of materials with higher melting points that may be useful to study for application to thermal-storage STP systems. For these materials, the heat of fusion should again be examined relative to the energy storage capacity of traditional batteries. The cost per kilogram of these materials has also been listed, as expense may be a concern in satellite system optimization; for most materials, however, the cost of launching an extra kilogram of material to LEO will likely far exceed the cost of the material itself. Table 2 lists some elements and compounds with melting points above 1500 K and heats of fusion at least 25% that of the energy storage density of a lithium-ion battery. As these are candidate phase-change TSMs, the maximum melting temperature has been limited to 3000 K so that the

TSM might be successfully contained within another material and an adequate means of heat exchange to and from the TSM can be designed.

A stand-out among the TSM candidates is boron. The melting temperature of boron, at 2350 K, is very close to the targeted temperature of 2500 K for optimal performance; at this temperature, an ammonia-fueled STP should achieve an Isp of over 390 s. Additionally, boron has a very high phase-change energy storage capacity of 4600 kJ/kg; this is an order of magnitude larger specific energy density than is available from state-of-the-art batteries today. The material cost per kilogram of boron is high relative to some of the other materials, but only a few thousand dollars worth of boron (~1 kg) would be required to provide a generous power budget on a microsatellite; a more realistic number to provide 1000 kJ of stored thermal energy would be 220 grams, costing $1100. There are some concerns regarding boron, however, in that it may be reactive with many high-temperature material candidates for use in the heat exchanger portions of an STP system; as noted later in this paper, several materials with melting points above 3000 K are boride compounds and several of those that are not borides contain elements which may react with boron to form borides in-situ[15]. Additionally, the thermal conductivity of boron is somewhat low; the effects of this trait would need to be investigated, as there are means to use a matrix of another material to ensure uniform temperature distribution and fast heat transfer when required[12,14].

Another notable TSM candidate from the table is common silicon dioxide. The melting point of SiO_2 is nearly 2000 K, and could potentially provide a solar thermal rocket with an Isp of over 350 s. Glass has a relatively low heat of fusion, so something on the order of 5 kg would be required to store 1000 kJ of thermal energy, eating into the weight budget of a microsatellite. Glass is, however, a common material to be melted and manipulated, so materials to contain it in that state are readily available and the design of a containment vessel and heat exchanger may prove significantly simpler than what would be required for a boron-based system. The low thermal conductivity of SiO2 is another engineering challenge that would likely need to worked to ensure uniform temperature and efficient transfer of heat to the propellant.

Roughly 4 kg of iron would be required as a TSM to provide 1000 kJ of stored energy. The melting point of iron, at 1811 K, would provide an Isp of 345 s. Iron, much like glass, is relatively inexpensive and is a common material to be melted and manipulated, likely simplifying the design process for a satellite. Provided the thermal conductivity of iron remained high as the temperature increased, utilizing iron as a high-temperature phase-change TSM could be relatively simple compared to some of the other options.

Another unique material with a notably high heat of fusion is silicon; with a heat of fusion of nearly 1800 kJ/kg, only 560 g would be required on board the spacecraft to store 1000 kJ. Silicon itself is very inexpensive, and is commonly melted and handled in the semiconductor industry, so this material could offer significant mass and cost savings if developed for a satellite. The melting point of silicon is 1687 K, which would provide an Isp over 330 seconds, which is still notably better than what can be attained with standard in-space chemical propulsion systems[1]. Silicon, at least at room temperature, has a high thermal conductivity, which could also lend benefits as a TSM.

THERMAL INSULATION MATERIALS

One critical technology for thermal energy storage/conversion systems is high temperature thermal insulation. Very high temperature thermal insulation materials must meet a long list of exacting requirements including the ability to withstand the storage temperatures, potentially operate through a number of thermal cycles, be compatible with the other involved materials, and maintain some structural properties. Some applications may also require that the material maintain a high transparency for a select radiation spectral band (either for incoming solar radiation or outgoing thermal radiation). The present work is focused on systems operating in the temperature range of 1500—2500K. There is a wide range of

materials available for that temperature range. The materials choice will be heavily dependant on the specific application, limiting the present discussion to only general considerations. Table 3 lists a select group of common high temperature thermal insulation materials along with a select group of other materials that represent a high-performance material for other select required properties. Aerogel is listed to illustrate the relevant properties for a material that is fully optimized for minimum thermal conductivity. It clearly can not meet the high temperature requirement, however. Similarly, fused silica is included for its optical properties, but it also can't achieve the required temperature. The effective thermal conductivity for blackbody radiation transport between the specified temperature and 300K for a gap distance of 1cm is also given for reference.

The mass of the required insulation can be estimated by assuming: a 1kg cube of boron is maintained at its melting point, 2350K, the cube is insulated from a cold side maintained at a temperature of 300K, the temperatures are maintained for a typical LEO eclipse period, 40 minutes, and the an allowable energy loss is 500kJ (approximately 10% of the energy available from the latent heat of fusion). An average thermal conductivity for the material is assumed. Carbon Bonded Carbon Fiber (CBCF) is a porous carbon fiber material that was used on NASA's general purpose heat source radioisotope thermal generator (GPHS-RTG) and it has attractive properties for the present discussion.[22] Using CBCF with an average thermal conductivity of 1.1 W/mK, the insulation mass for the assumed conditions would be 1.4kg. This is clearly within the mass budget for the representative microsatellite system under discussion, indicating that a material with properties similar to CBCF is required for the insulation of the system. If an optically transparent material such as silicon carbide (average k_{th} ~ 25W/mK) was employed it would require 840kg of material. This simple analysis has indicated that that state-of-the-art opaque thermal insulation materials are strong candidates for application, but that optically transparent materials with the required properties will need additional development. It is also worth noting that the blackbody radiation emitted from the cube would amount to 1.4MW/m^2 of radiant energy. For this system (A = 3.3x10^{-2}m^2 , Δt=2400s), this would amount to 56kW or 135MJ in 40 minutes which is insufficient for application. Radiation shielding along with carefully designed supports would be required for vacuum insulation to be viable.

Table 3: High Temperature Insulation Properties

Material	Clear	Density [kg/m^3]	T_{melt} [K]	$k_{th,500K}$ [W/mK]	$k_{th,1000K}$ [W/mK]	$k_{th,1500K}$ [W/mK]	$k_{th,2000K}$ [W/mK]	$k_{th,2500K}$ [W/mK]
Aerogel[16]	Some	80	600	0.01				
Fused Silica[19]	Yes	2200	1985	1.5	2.1	2.1		
Saphire[18]	Yes	4000	2313	20	8	---	---	
Alumina[18]	No	4000	2345	21	5	5	8	
Boron Carbide[17]	No	2520	2673	12.5	9	6.5	---	---
Silicon Carbide[21]	Yes	3210	3003	120	60	38	28	---
Boron Nitride[18]	Some	3487	3246	37	22	21	19	---
Carbon Bonded Carbon Fiber[20]	No	180	3273	---	0.4	---	0.9	---
Vacuum [Δx=1cm]	Yes	---	---	0.15	0.80	2.39	5.33	10.1

TSM CONTAINMENT AND HEAT TRANSFER TO PROPELLANT

It has been shown in the literature that on similarly sized satellites, power delivered by a solar thermal system (via heat collection and transfer to propellant) greatly exceeds that achieved on an electrically powered propulsion systems. A variety of geometries have been designed and tested to maximize the heat transfer to the propellant[2,13,23]. Various means to enhance heat transfer by containing the thermal storage medium in a conductive matrix[12,14] or by utilizing natural convection[13] have also been explored. In designing a solar thermal system, therefore, the primary focus of optimizing the heat transfer to the propellant will be on the materials and design elements required to handle the 2500 K target temperature for propulsion.

Materials with melting temperature above 2500 K that may be useful for heat exchange and/or containing the TSM are listed in Table 4. For the most part, materials with such high melting temperatures are limited to transition metals, especially those in groups 4 and 5 on the periodic table, along with their oxides, carbides, borides and nitrides[15]. Oxides may be of limited use, however, as they typically have poor thermal shock resistance and high thermal expansion, creating difficult engineering problems for a solar thermal system. Carbon represents the element with the highest melting point, and has one of the highest melting points of all known materials. Rhenium, in particular, has been noted as a good candidate for use in solar thermal systems due to its ductility and weldability while also having a superior ability to withstand high temperatures[2]. Graphite is commonly used in the heat-exchange portion of STP systems, though it may require a coating of Rhenium to prevent harmful interactions with hydrogen at high temperatures[2]. The monocarbides of tantalum and hafnium are noted as being particular useful in applications where temperature rises quickly to levels above 2275 K, and so may be of use for STP applications[15].

For many of the high temperature compounds, properties can be tailored by adjusting the stoichiometry slightly or by mixing with small amounts of other compounds and elements; $HfC_{0.67}$, for example, has been shown to be less susceptible to damaging oxidation than $HfC_{0.98}$ [15]. Additionally, certain elements and compounds can benefit from a thin coating of glass oxide. HfO_2, for example, has a significantly higher melting temperature than pure hafnium; allowing a thin oxide layer to form can increase the durability of the hafnium in certain environments while maintaining the bulk properties of hafnium itself[15].

For establishing heat transfer between the TSM and the propellant, a material with a high thermal conductivity may be required as the heat exchanger material. Molybdenum has a thermal conductivity of 138 W/mK, while rheniums is much lower at 39.6 W/mK[2]. Zirconium Boride (ZrB_2) and Hafnium Boride (HfB_2) are both noted to have room-temperature thermal conductivities approaching that of copper[15], or roughly 400 W/mK. It should be noted, however, that establishing a highly conductive heat exchanger that does not bleed off significant thermal energy when propulsion is not required will pose a significant design challenge; the heat exchanger and the attached propellant feed system should not be a path for heat transfer through the insulator used to maintain TSM temperature. While adequate heat exchangers, including radiation shields and other structures, have been designed for traditional STP systems[5], the addition of thermal storage produces additional challenges.

THERMAL-TO-ELECTRIC CONVERSION SYSTEMS

As discussed earlier, the most appealing thermal power system for microsatellite applications is a hybrid system that operates as both the propulsion system and the electrical power system. A wide variety of thermal-to-electric power conversion systems have been proposed/developed, including both static systems (thermoelectric, thermophotovoltaic, thermionic, alkali metal thermal-to-electric conversion (AMTEC), and nantennas) and dynamic (primarily closed Brayton cycle and free piston Stirling cycle). These systems have been thoroughly reviewed elsewhere and will only be briefly discussed here.[24-27]

Table 4: Materials with melting temperatures above 2500 K [4,15]; some property and cost figures have been gleaned from internet sources and supplier specification lists and direct references are not provided here. Thermal conductivities are noted at room temperature.

Element/Compound	Atomic Number	Melting Temp [K]	Thermal Conductivity [W/mK]	Cost [/kg]
Hafnium	72	2506	23.2	$1,200
Ruthenium	44	2607	117	$14,000
Iridium	77	2739	147	$42,000
Niobium	41	2750	53.7	$170
Molybdenum	42	2896	138	$100
ZrO_2	N/A	2988		
SiC	N/A	3003		
HfO_2	N/A	3031		
WC	N/A	3143		
NbB_2	N/A	3173		
BN	N/A	3246		
ZrN	N/A	3250		
Tantalum	73	3290	57.5	$200
Osmium	76	3306	87.6	$77,000
TaB_2	N/A	3310		
TaN	N/A	3363		
TiC	N/A	3433		
Rhenium	75	3459	39.6	$16,000
TiB_2	N/A	3503		
HfB_2	N/A	3520	~400	
ZrB_2	N/A	3520	~400	
HfN	N/A	3580		
Tungsten	74	3695	173	$110
NbC (or Nb_2C)	N/A	3763		
ZrC	N/A	3800		
C (diamond)	12	3820	470	variable
C (graphite)	12	3948	25-470	$5
TaC (stoichiometric)	N/A	4150		
HfC	N/A	4200		
$TaC_{0.89}$	N/A	4270		
Ta_4HfC_5	N/A	4488		

Thermal-to-electric conversion systems for microsatellites must meet a variety of performance and operational metrics. The system level specific power (We/kg) is usually the most critical and will be the focus of this discussion. One other metric, the conversion efficiency, will also be discussed because of its effect on specific power. The limiting energy conversion efficiency for all heat engines operating between T_h and T_c is the Carnot efficiency:

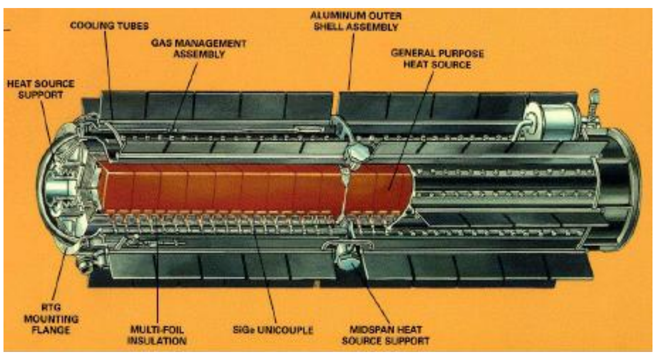
$$\eta_c = 1 - \frac{T_c}{T_h}. \quad (2)$$

Figure 3 includes a plot of η_c for a heat engine operating between T_h and T_c with T_c held at 300K. The Carnot efficiency for a system operating with T_h between 1500K and 2500K is between 80% and 88%.

Further increasing T_h beyond 2500K could increase the thermal energy storage density (likely at decreased system lifetime), but will have only modest effects on the maximum theoretical efficiency of the system.

EXISTING STATE-OF-THE-ART: GENERAL PURPOSE HEAT SOURCE (GPHS)

The United States has flown satellite thermal-to-electric conversion systems in the form of radioisotope thermal generators (RTGs), with the General Purpose Heat Source Radioisotope Thermal Generator (GPHS-RTG) developed for NASA deep space missions representing the current state of the art.[22] The GPHS will be used as a starting point for the discussion in this section. There are two primary differences between the GPHS and hybrid power/propulsion systems: the energy storage media ($_{238}$Pu for the GPHS) and the hot-side operating temperature (1273K for the GPHS). The GPHS uses a SiGe thermoelectric conversion system because of its competitive specific power, robustness, and long lifetime. A schematic of the GPHS and its pertinent performance parameters are given in Figure 2. The specific power, without energy storage, for the GPHS-RTG is 9.4 We/kg. For the system discussed in this work, that would correspond to a mass of 10.6kg, which is marginally acceptable.



P_{out} : P_{in} [W] (BOM)	285 : 4500
η_e : η_e/η_c	6.3% : 14%
System Mass [kg]	56.0
$m_{storage}$: $m_{conversion}$ [kg]	25.7 : 6.2
$m_{cooling}$: $m_{insulation}$ [kg]	13.0 : 6.4
Specific Power [We/kg]	5.1
T_h,T_c [K]	1273, 573

Figure 2. The GPHS-RTG[22,26]

STATIC THERMAL-TO-ELECTRIC ENERGY CONVERSION SYSTEMS

The microsatellite thermal energy systems viability of five static conversion systems will be discussed in this section: thermoelectric, thermophotovoltaic, thermionic, AMTEC, and nantenna. The current state of the art spacecraft thermal energy systems use thermoelectric conversion despite their relatively low conversion efficiencies (<10%) and specific power levels (~10We/kg) because of their proven reliability. The thermoelectric effect occurs when a temperature gradient across a thermoelectric material causes charge carrier diffusion via multiple mechanisms which, in turn, generates a potential difference across the material. The fundamental performance metric for thermoelectric systems is their figure of merit, ZT. Figure 3 shows the efficiency of the thermoelectric conversion process for representative ZT values. Table 5 lists properties of commonly used thermoelectric materials.

Recent efforts with quantum dot materials and nanostructured materials have indicated that higher ZTs, and conversion efficiencies, are possible.[28] It is reasonable to assume that steady improvements in GPHS-like systems will achieve the desired conversion specific mass (not including energy storage) of 10We/kg. The primary difficulty with their application, however, is operating at the high temperatures that are being discussed (1500K - 2500K).

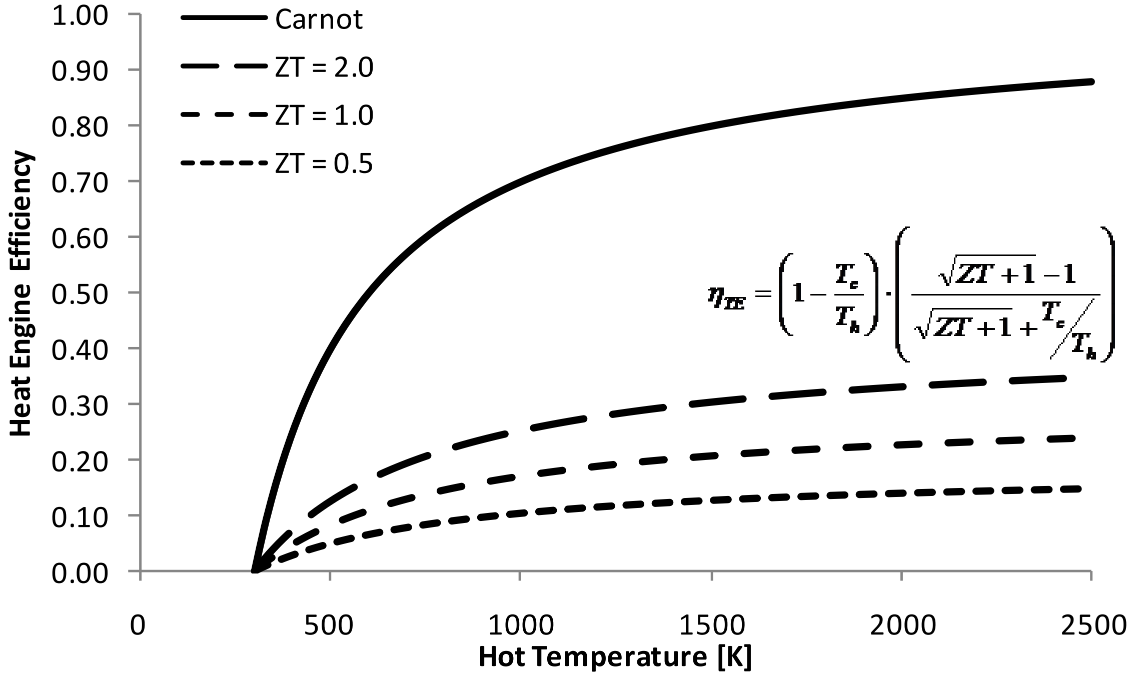


Figure 3. Thermoelectric Electric Conversion Efficiency

Table 5. Common Thermoelectric Materials[28]

Thermoelectric Material	Maximum Temperature [K]	Max Figure of Merit (ZT)
Bi_2Te_3/Sb_2Te_3	475	1.0 (n), 1.0 (p)
PbTe/TAGS	800	0.7 (n), 1.2 (p)
SiGe	1273	1.0 (n), 0.6 (p)

Thermophotovoltaic (TPV) systems are also under development for space-based radioisotope power sources.[24,29,30] TPV energy conversion systems operate on the same principle as solar photovoltaics, but have lower energy band gaps that correspond to the peak emission of blackbodies of around 1275K - 1575K. TPV systems are broadly separated into two classes: broadband radiator with complex high performance filter/converter, or narrowband emitters with low-cost silicon converters. Creare has demonstrated a system for NASA with a conversion efficiency of 19% and a projected systems level specific power of greater than 12 We/kg (including energy storage).[26] Operating a TPV system for microsatellite thermal energy systems may require a different choice of materials to properly match the materials' bandgap to the peak radiation wavelength, but the technology appears viable as a mid-term technology.

Thermionic thermal-to-electric conversion systems consists of a hot cathode which thermionically emits electrons over a potential energy barrier to a cooler anode to produce electrical power. They have been studied for use in space nuclear reactor systems.[24] The majority of investigated systems use cesium vapor in the collisional regime. The theory behind their operation is relatively simple, but reducing it to practical devices has proven difficult. For space power systems, the emitter temperature is between 1700K and 2100K and the collector temperature is between 700 - 1000K. Because of the high

temperatures, almost all thermionic electrodes are refractory metals. A thermionic conversion system was baselined for the SOTV mission which was designed to operate with a hot-side temperature of 2200K.[8] Thermionic conversion systems, however, tend to be more advantageous at higher power levels and may not be scalable to microsatellites.

The alkali metal thermoelectric converter (AMTEC) drives sodium around a closed thermodynamic cycle between two heat reservoirs at different temperatures.[24] The cycle operates by the selective ionic conductivity of Beta-alumina solid electrolyte (BASE), which causes an isothermal expansion of sodium vapor through a solid electrolyte, converting the work of isothermal expansion directly into electrical power. AMTEC systems currently under investigation operate with cold side temperatures of 400 - 700K and hot-side temperatures of 900K-1300K. Efficiencies of 16% have been demonstrated in the lab with efficiencies as high as 35-40% predicted with future developments. The ultimate temperature limitation of the technology is limited to the melting point of BASE (2253K) and the material compatibility of BASE with the working fluid (Na: 1,300K). A system is predicted to have a total solar thermal systems specific power of 14We/kg. The limited temperature of operation appears to be the primary limit to the technology.

A recently investigated far-term static conversion technology is arrays of nantennas.[27] A nantenna is a nano-antenna that is used to convert incoming solar/thermal electromagnetic radiation into electrical energy. Idaho National Labs has designed a nantenna to absorb in the wavelength range of 3-15 μm (infrared). However, several problems are encountered when scaling conventional scale antennas to work at vary small scales and high frequencies. Very small antennas carry almost all of the electrical current on the surface, which leads to fundamentally different operation. The primary difficulty with the technology, however, is manufacturing the required rectifying diode to operate at the extremely high frequencies at which nantennas must operate (100s of THz). Single wavelength absorption efficiencies exceeding 90% have been demonstrated, but the efficiency of complete systems operating with a distribution of wavelengths and the inadequate rectifying diodes has not reached 1%. Nantennas are an exciting technology with significant long-term potential, but it is impossible to make accurate projections of the viability of nantennas for spacecraft thermal-to-electric conversion systems until the technology is further advanced.

DYNAMIC THERMAL-TO-ELECTRIC CONVERSION SYSTEMS

Dynamic conversion systems generally operate at higher conversion efficiencies and scale better for high power levels, but suffer from concerns over the long-term reliability of the moving parts.[24] No complete system has yet been flown in space for the United States. There are a large number of thermodynamic cycles that could be used in spacecraft thermal power systems, but two are most common: the closed-Brayton cycle and the free piston Stirling cycle. The Brayton cycle is the thermodynamic cycle used in turbines: isentropic compression, isobaric heat addition, isentropic expansion, isobaric heat rejection. Space based Brayton cycle conversion systems have been under development with power levels ranging from 500We to 100kWe since the early 1960s. A system efficiency of η=29% has been demonstrated for a 2kW system using relatively old technology.[31] Closed-Brayton cycle systems do not scale linearly with power and are typically most competitive at very large scales. Axial turbine inlet temperatures are limited to approximately 1150K for superalloys and 1450K for refractory metals (susceptible to corrosion).[24] Radial turbines, however, are typically better performing at low power levels and may achieve T_h=1700K with ceramic versions or above 2000K with carbon-carbon composite versions. Radial closed-Brayton cycle conversion systems may become applicable for the microsatellite applications discussed in this paper if the required materials and manufacturing techniques can be developed for carbon-carbon composite systems.

The free piston Stirling cycle is an efficient and simple system that is typically lightweight and doesn't require lubricants or high-pressure seals. Sunpower is developing a small-scale Stirling engine that achieves over 50% of Carnot efficiency with 100We/kg.[32] Free piston Stirling engines have also

demonstrated lifetimes of over 60,000 hours. The primary concern with free-piston Stirling engines is that they are externally heated, requiring several parts of the device to be constantly exposed to high temperatures. A current state-of-the-art system, the SSPC, demonstrated operation at 1050K, which is considered the limit for superalloys.[24] A significant effort would be required to go beyond that limit, indicating that free-piston Stirling engines are not a good choice for high-temperature thermal energy conversion systems for the near future.

Table 6. Comparison of Thermal-to-Electric Conversion Technologies

Technology	P_{sp} [W/kg]	Efficiency	T_{max} [K]	Comments
Thermoelectric	9.4	6.3%	1273	• Limited temperature operation. • Incremental development OK for P_{sp} & η.
Thermophotovoltaic	15	19%	None	• Long lifetime demonstration required. • Operation in space environment required.
Thermionic	100@1kWe	> 10%	2200	• Baselined for another application at higher power levels.
AMTEC	14	16%	1300	• Unlikely to achieve required temperatures.
Nantenna	???	<1%	None	• Significant uncertainties in all aspects. • Current concern: efficient rectifying diode.
Closed Brayton	---	29%	1700	• Possible to achieve required temperatures. • Mass may be high for low power levels.
Free Piston Stirling	100	35%	1050	• Unlikely to achieve required temperatures. • High performance & long lifetime demonstrated.

<u>SUMMARY</u>

There are a wide variety of options for thermal-to-electric systems that operate at a T_h below 1300K and are not mass constrained. Several of the options appear fundamentally constrained on T_h and would require significant development efforts to achieve the required T_h for high-temperature thermal-to-electric conversion on spacecraft: thermoelectric, AMTEC, and free piston Stirling cycle. Closed-Brayton cycle systems that can operate at the required high temperatures are not currently available, but it is likely that with significant development the technology may be advanced significantly with carbon-carbon systems. They are, however, typically the most advantageous for large power levels. Thermionic power conversion systems are also typically more advantageous at higher power levels. Nantenna based conversion systems have significant potential, but have yet to achieve any significant fraction of their potential. That leaves thermophotovoltaic systems as the most promising candidate for the near term. TPV systems will be chosen as the representative solution for the following discussions. Table 6 shows a

summary of the comparison of the state-of-the-art of different energy conversion technologies and comments about the future potential of the technology.

SYSTEM DESIGN SUGGESTIONS

ENERGY AND MASS BUDGETS

The thermal storage insulation has been designed such that a maximum of 500 kJ is lost during eclipse. For propulsion during eclipse, we assume a maximum continuous (or average) thermal power draw of 100 Watts, totaling 240 kJ. A net total of 240 kJ of usable energy has also been budgeted to the electrical system during each eclipse cycle; with a 20% conversion efficiency as may be achieved with thermionic or thermophotovoltaic systems, this requires 1200 kJ of stored thermal energy. Therefore, the total thermal storage for a highly capable microsatellite is roughly 2000 kJ, which can readily be stored with 0.5 kg boron. With 1.4 kg of insulation, a total of less than 2 kg would be required for the thermal storage component, yielding a net specific energy density of 1000 kJ/kg, or double that of a top of the line battery.

While illuminated by the sun, the satellite will have 60 minutes to store up 1200 kJ, while allowing the possibility of simultaneously providing surplus energy for 100 Watts of propulsion and 100 Watts of electrical power (500 Watts delivered to a 20% efficient conversion system). Based on earlier calculation targets for insulation of the TSM, 750 kJ would be lost via radiation during this 60-minute period. So, during each 60-minute period of insolation, a total of 4100 kJ must be delivered to the TSM. The delivered concentrated solar power must therefore be 1150 Watts. Assuming a 50% efficient concentration and collection system (an efficiency that should be exceeded in a well designed system[5]), this could be achieved with less than 2 square meters of solar concentrator, weighing less than 2 kg [2].

With the analysis above, the solar concentration, collection, and thermal storage system could come in at less than 5-10 kg once fiber optics, the propellant feed system and heat exchanger, and structural supports are included. The use of an ammonia STP system yields significant mass savings relative to conventional rockets due to the high Isp offered relative to chemical systems, so the mass of stored propellant will be relatively small when compared to that required for a chemical rocket with the same DV capability. A large contribution to spacecraft mass will come, however, from the relatively low specific power offered by many of the thermal-electric conversion technologies. For a system that can produce 100 Watts output, this could range from 1 kg for a free-piston Stirling engine up to 11 kg for a thermoelectric system.

With the above considerations, and assuming an Isp approaching 400 seconds and the highly ambitious DV capability of 1.5 km/s from an earlier study[1], the required propellant mass for a 100 kg microsatellite would be in the 40-50 kg range. Considering a worst-case summation of the component masses for this highly capable microsatellite (1.5 km/s DV capability, with full-time 100 Watt power capability and 100 Watt power to the propellant), over 25 kg would still be available for the payload. With a more typical microsatellite DV capability, the payload fraction would increase significantly.

SYSTEM INTEGRATION

Of the systems required for the spacecraft design proposed here, solar collection and concentration scheme have already been devised and discussed in the literature; likewise, there is no need to further discuss the propellant storage and feed systems or nozzles that might be involved. In the system design discussed here, the most complicated aspects will come in interfacing with the TSM such that energy can be added or extracted without significant losses. Therefore, this section will discuss starting-point designs for the heat delivery system, the propellant heat exchanger, and the electrical conversion system.

Heat delivery to TSM: At the 2350 K melting temperature of boron, radiated power for a black body (σT^4, where sigma is the Steffan-Boltzmann constant) is 1.73 MW/m^2; note that this ignores the fractional

emissivity of a non-black body emitter). With a concentration of 10,000 suns, assuming 50% transmission efficiency from the concentrated spot to the TSM and a solar flux of 1350 W/m^2, the intensity of light reaching the TSM should be 6.8 MW/m^2. It is possible, therefore, to provide a net input of light by providing optical access for the concentrated solar illumination to reach the TSM, even while radiated power escapes by the same path; this is especially true if the 50% transmission/delivery efficiency can be improved to the levels expected[5]. Using a high temperature lens (perhaps diamond) to focus the light emitted from the fiber optic cables through a pinhole could further multiply the favorable power balance. Shaping of the surface of the TSM (or its container) can also be used to limit the leakage of radiated power. The geometry should be adjusted to most effectively trap incoming radiation and limit outgoing losses; an approximation of such a geometry is illustrated in Figure 4. An optical switch may also be beneficial to reflect radiated power back into the TSM cavity and preserve energy while the craft is in eclipse. It is also worth noting that fiber optic cables that can operate at nearly 1200 K have been noted in the literature[5], so only minimal shielding should be required for their protection.

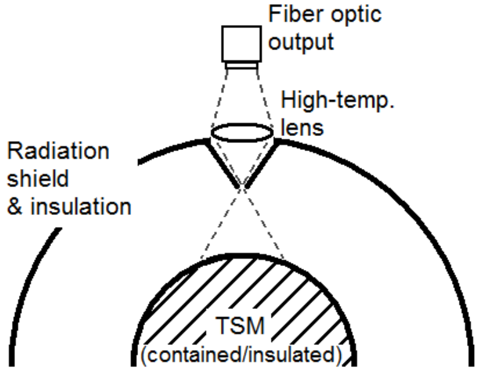


Figure 4: Illustration of simple radiation trap for adding concentrated sunlight to the thermal storage material.

Propellant Heat Exchanger: For the propellant heat exchanger, the primary focus of design should be on maximizing thermal contact and conduction between the TSM and the propellant, while minimizing any conductive losses that might occur along the heat exchanger (or the propellant feed connections) to other parts of the spacecraft. This may involve using insulating components to connect the propellant feed system to the heat exchanger within the TSM. If the thermal conductivity of the TSM is low, the system may also benefit from a finned heat exchanger, perhaps constructed of graphite, that can more rapidly draw heat from the TSM. Some benefit may also be gained by adjusting the overall geometry of the TSM storage system. For example, a cylindrical storage cell may allow for a relatively long heat exchanger along the cylindrical axis with relatively short fins extending radially to enhance heat transfer. A starting point design is illustrated in the Figure 5.

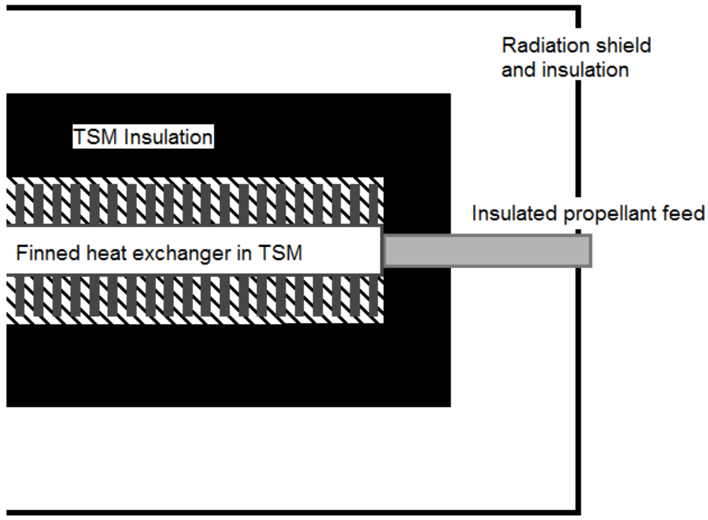


Figure 5: **Illustration of simple heat exchanger designed to minimize heat losses and maximize heat transfer with TSM.**

Thermal-to-Electric Energy Conversion: The nominal system requires a constant power level of 100W. For a 20% efficient conversion system, a total of 500We must strike the useable area of the TPV system. An ideal blackbody at the melting temperature or boron (2350K) would emit a radiant flux of 1.7MWe/m^2. With a relatively conservative estimate of a 50% reduction due to non-ideal emissivity and transmission losses, this yields a radiant flux of 0.86MWe/m^2 at the conversion surface. An emitting surface area of approximately 0.6cm^2 is required to achieve the required electrical power level for thermal-to-electric conversion. It should be noted that the collection area may need to be significantly larger depending on the maximum flux capabilities of the chosen technology.

SUMMARY AND CONCLUSIONS

A solar thermal propulsion system augmented with thermal storage and thermal-to-electric conversion has been suggested and analyzed. Past research into solar thermal propulsion has shown that, while the thrusters themselves offer significant advantages over conventional chemical rockets, the non-conventional (i.e.: non-chemical and non-electric) nature of these rockets lends its own drawbacks; these include a lack of augmented propulsion during eclipse and competition with the conventional power systems required for other spacecraft systems. It has been shown here that with a thermal storage system, particularly one utilizing a phase-change material, propulsion during eclipse can be optimized and additional advantages in terms of system mass and efficiency can be achieved. By coupling the thermal storage system with a thermal-to-electric conversion system and eliminating the conventional solar panels and batteries from a satellite, the apparent trade-offs for a solar thermal propulsion system can be mitigated, and a highly functional, highly capable satellite can be produced.

A first-level analysis has been completed for the assorted subsystems of such a satellite, including solar collection and concentration, heat delivery to and insulation of the thermal storage medium, heat exchange to the propellant, and thermal-to-electric conversion. There do not appear to be major breakthroughs or advancements in materials required to achieve the desired performance; materials of the desired thermal storage density and insulation level exist and are reasonably attainable. The remaining difficulties in assembling a satellite like that described here will entail insuring material compatibilities at high temperature, and assembling the various subsystems in such a way that no component detracts significantly from the performance of the other subsystems.

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